# Carbon-Carbon Radiator Validation Report

**February 13, 2002**

Nicholas M. Teti
*Swales Aerospace*
*Beltsville, Maryland 20705*

**NASA/GSFC**

## Table of Contents



## List of Illustrations



## List of Tables

# 1. INTRODUCTION

The New Millennium Program (NMP) Earth Observing-1 (EO-1) spacecraft uses six passive radiators, each consisting of sandwich panels constructed with aluminum honeycomb core. Five of the six radiator panels use standard aluminum facesheets. On the sixth radiator panel, as an EO-1 technology demonstration item, the aluminum facesheets were replaced with an experimental panel that utilized Carbon-Carbon (C-C) material for its facesheets. The objective of this technology validation was to demonstrate that C-C could be a cost efficient facesheet material for honeycomb core radiator panels that also function as part of the primary spacecraft structure. In this case the radiator serves as a shear panel.

The Carbon-Carbon Radiator (CCR) panel is a 28.62 in. x 28.25 in. sandwich composite panel with two 0.022-in-thick C-C facesheets bonded to a 1-in 5056 aluminum honeycomb core with a density of 2.1 lb/ft$^3$ weighing approximately 5.5 lb. The facesheets of the panel are made of carbon fibers in a carbon matrix. The internal surface of the CCR panel is coated with an epoxy encapsulate (Figure 1) to prevent particle contamination of sensitive instruments on board EO-1 and provide additional strength to the panel. The external surface (Figure 2) of the CCR panel is coated with silver Teflon as required by the EO-1 spacecraft thermal design.

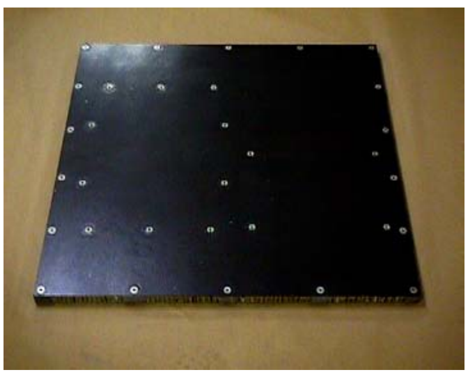

**Figure 1. Carbon-Carbon Panel Internal Surface**

**Figure 2. Carbon-Carbon Panel External Surface**

The use of high conductivity fibers in C-C fabrication yields composite materials that have high stiffness and high thermal conductivity. Since C-C density is lower than that of aluminum, significant weight savings may be realized by replacing aluminum panels with such panels. A C-C material also has an advantage over other high conductivity composite materials in that the thermal conductivity through the thickness is considerably higher. The trend for future satellites is towards higher power density in combination with a reduction in spacecraft size and weight. Since C-C materials also have a markedly higher specific thermal efficiency than aluminum, they offer improved performance for lower volume and mass. They will enable more compact packaging of electronic devices because of their ability to effectively dissipate heat from high power density electronics. Studies have shown that entire heat pipe panels may be replaced by high-conductivity C-C for some applications, thus reducing system complexity as well as integration and testing costs. Also, since C-C is a structural material, it may serve a dual purpose as both a structural and a thermal management material. The material may eventually eliminate the need for thermal doubler plates, which typically add substantial mass to a spacecraft. Finally, because C-C is a composite, its structural and thermal properties are tailorable, thus adding capability and flexibility to spacecraft designs.

Due to high fabrication cost and low interlaminar shear strength, successful application of C-C composite materials have been limited to non-structural thermal protection and frictional applications. Examples include the space shuttle wing leading edges and aircraft brakes. However, recent advances in C-C materials fabrication have improved the viability of C-C for thermally demanding structural applications.

The panel was built by the Carbon-Carbon Spacecraft Radiator Partnership (CSRP). The CSRP, with the participants shown in Figure 3, was an informal partnership established to promote the use of C-C on spacecraft. To demonstrate the structural potential of C-C, the CSRP built a structural panel and subjected the sub-components (facesheets and insert regions) as well as the integrated panel to numerous tests.

**Figure 3. CSRP**

The CSRP was started by Howard Maahs of NASA Langley Research Center and Elizabeth Shinn of Wright Patterson Air Force Base. The CRSP quickly grew to include an informal partnership with additional members from NASA Goddard Space Flight Center, the Naval Surface Warfare Center, TRW, Lockheed Martin, Amoco, BF Goodrich, Materials Research & Design, and Swales Aerospace. Representatives from the CSRP, as shown in Figure 3 from left to right, are as follows: Braford Parker (NASA/GSFC), Suraj Rawl (LM-AD), Joe Wright (LM-V), Dan Butler (NASA/GSFC), Eric Becker (AFRL/MLBC), Brian Sullivan (MR&D), Wallace Vaughn (NASA/LaRC), Elizabeth Shinn (AFRL/MLBC), Howard Maahs (NASA/LaRC), Al Bertram (NSWCCD) and Steve Benner (NASA/GSFC). Not present were Ed Silverman (TRW), Jim Findley and Andy Klavins (LM-M&S), Cris Sprague (Amoco), Wei Shih (BF Goodrich), Waylon Gammill, Don Gluck, and Charlotte Gerhart (AFRL/VSVT).

## 2. TECHNOLOGY DESCRIPTION

Carbon-Carbon is a special class of composite materials in which both the reinforcing fibers and matrix materials are made of pure carbon. Mechanical properties of the particular C-C material used for the CCR facesheets are given in Table 1. The use of high conductivity fibers in C-C fabrication yields composite materials that have high stiffness and high thermal conductivity. The primary thermal function of the EO-

1 CCR is to radiate the 27.8 watts generated by the EO-1 Power Supply Electronics (PSE) and the 16.3 watts (peak power) generated by Linear Etalon Imaging Spectral Array/Atmospheric Corrector (LEISA/AC) electronics boxes. The panel also serves as a primary structural member and must support the combined weight of the PSE (50 lb) and the LEISA/AC (10 lb) boxes while being subjected to severe mechanical launch and thermal on-orbit environmental loading conditions. Figure 4 shows the comparative differences between the CCR and a baseline radiator design.

**Table 1. Carbon-Carbon Composite Facesheet Mechanical Properties**

| Material Properties | | Material Allowables | |
|---|---|---|---|
| $E_1 = E_2 =$ | 1.62e7 psi | $S_1^t = S_2^t =$ | 30.7e3 psi |
| $G_{12} =$ | 6.11e6 psi | $S_3^t =$ | 1.0e3 psi |
| $G_{1z} = G_{2z} =$ | 0.35e6 psi | $S_1^c = S_2^c =$ | 12.7e3 psi |
| $\nu_{21} = \nu_{12} =$ | 0.32 | $S_3^c =$ | 6.0e3 psi |
| $\alpha_1 = \alpha_2 =$ | -1.182e-6 1/(°C) | $T_{23} = T_{13} =$ | 1.4e3 psi |
| | | $T_{12} =$ | 10.0e3 psi |

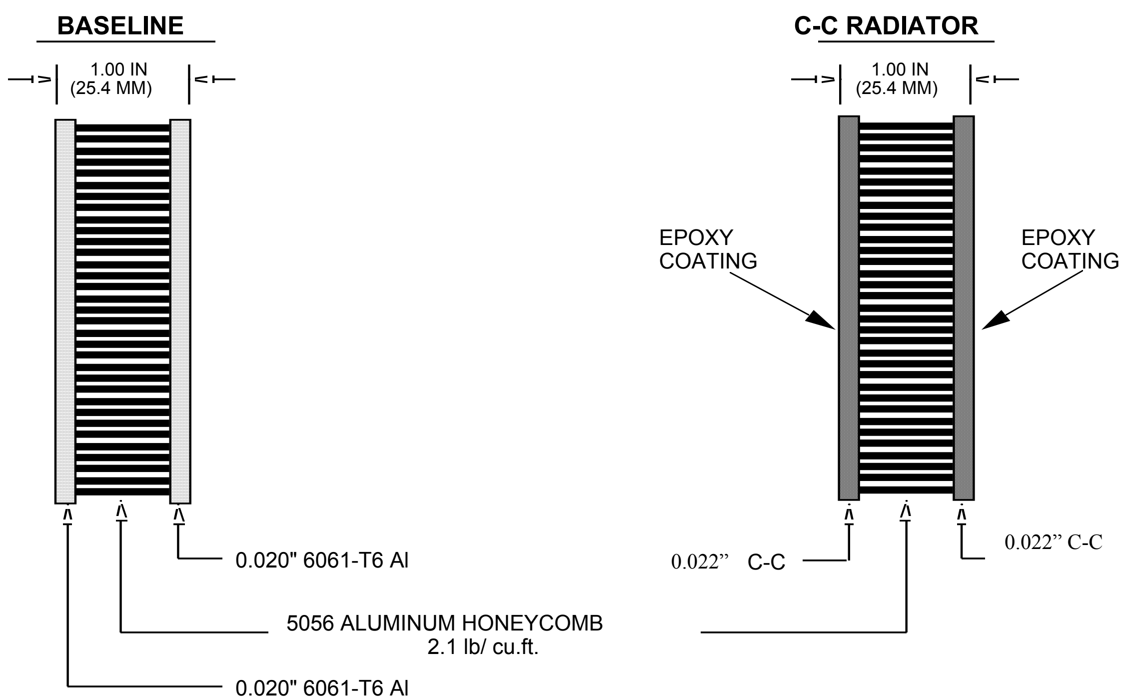


**Figure 4. Carbon-Carbon and Baseline Radiators Cross-Sections**

The panel is attached to the EO-1 spacecraft bus through 18 attachment points at the perimeter and supports the two electronics boxes through 14 attachment points on the interior with eight additional inserts to support EO-1 GSE. The CSRP selected the C-C face sheet design based on a material trade study by Materials Research and Design. Each face sheet is composed of two plies of 5-harness stain weave fabric as shown in Figure 5. The fabric is constructed from P30X carbon fibers and the carbon matrix is introduced by Chemical Vapor Infiltration (CVI). BF Goodrich fabricated the face sheets and Lockheed Martin Vought Systems designed the perimeter and interior attachment point configurations. The design consists of aluminum inserts bonded to the honeycomb core and facesheets with potting compound as shown in Figure 6. The perimeter insert design used to attach the panel to the spacecraft is shown in Figure 6a. The interior insert design used to attach the electronics boxes is shown in Figure 6b. In both cases, the potted insert design is such that the inserts do not bear directly against the facesheets. Load is transferred through the inserts to the potting compound and into the facesheets and honeycomb core. In this configuration the potting compound effectively acts like an elastic spring between the facesheets and spacecraft frame to relieve thermal expansion mismatch stresses.

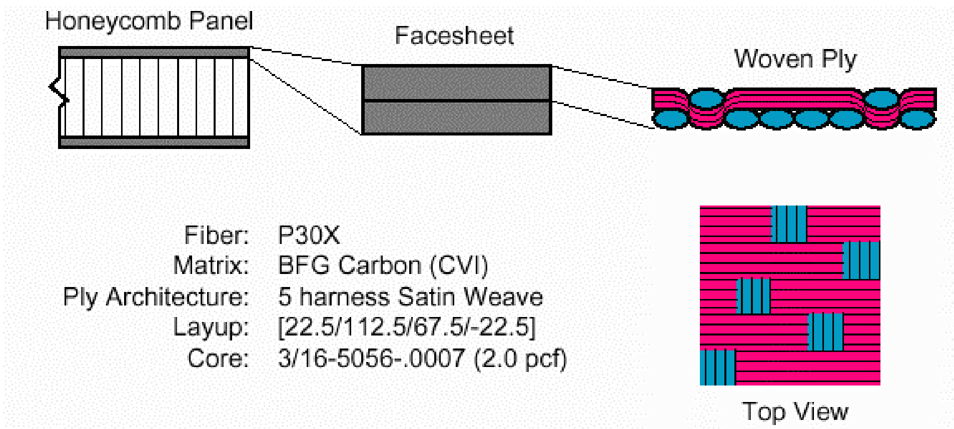


**Figure 5. Carbon-Carbon Radiator Facesheet Construction**

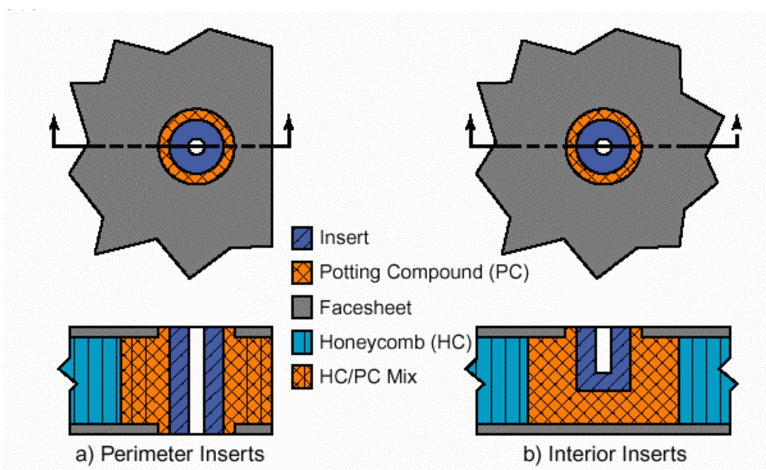


**Figure 6. Carbon-Carbon Radiator Panel Insert Design**

Clearance through-hole diameters in the perimeter inserts were specified by Swales Aerospace to reduce loads induced the by thermal mismatch between the aluminum spacecraft and C-C panel.

Further background information that describes the CCR technology and pre-launch testing can be found in References 1 and 2.

# 3. TECHNOLOGY VALIDATION

The EO-1 image shown in Figure 7 shows the CCR panel in its final configuration on the launch vehicle at Vandenberg, AFB. The Silver Teflon radiator area is 12.0 in. x 16.25 in.. The remaining external surface is covered with MLI having a 3-mil Kapton outer layer.

**Figure 7. EO-1 Prior to Launch**

The flight configuration CCR panel was initially instrumented with six (6) thermistors on the internal face sheet and one (1) thermistor (TBAY4T) on the front/external (space viewing) face sheet. Figure 8 below shows the thermistor locations and telemetry mnemonics. However, as part of the plan to accommodate the EO-1 calorimeters, one of the internal thermistors was removed as noted.

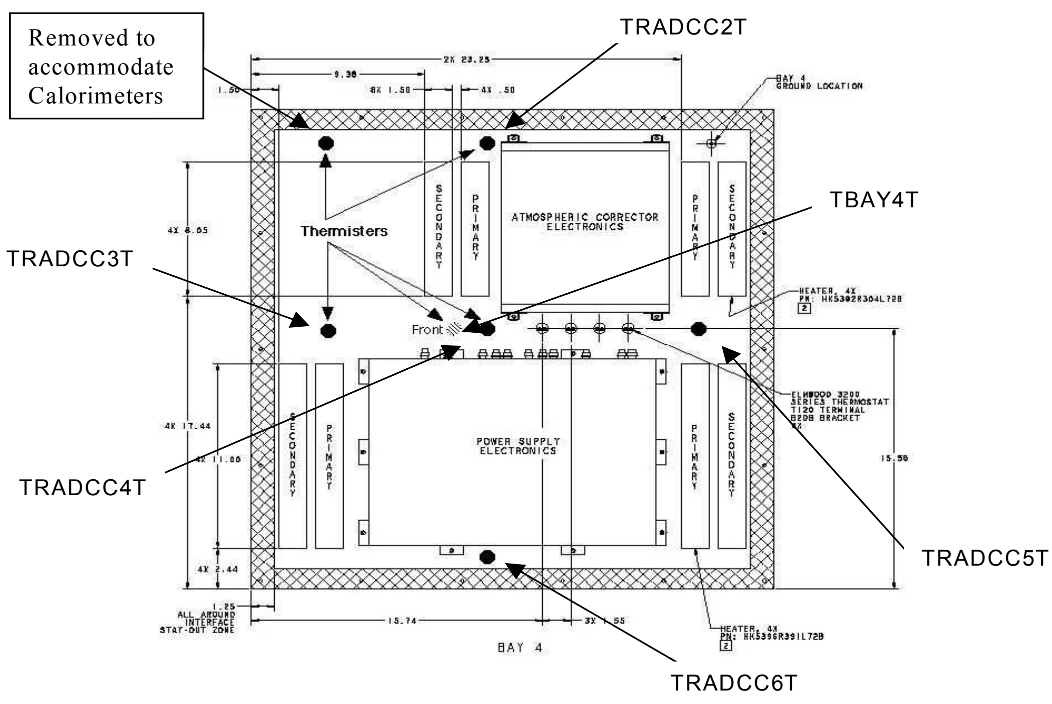


**Figure 8. Carbon-Carbon Radiator Panel Thermistor Locations**

The CCR panel replaces one of the six structural panels on the EO-1 spacecraft (Bay 4 panel). Simplified geometric math models (GMMs) and thermal math model (TMM) of the panel were developed from the detailed EO-1 spacecraft model for this analysis effort. The GMMs are Thermal Synthesis System (TSS) models (Figure 9) and the TMM is a Systems Improved Numerical Differencing Analyzer (SINDA) model. The EO-1 early on-orbit spacecraft model correlation was excellent and thereby provided a good basis for this correlation effort.

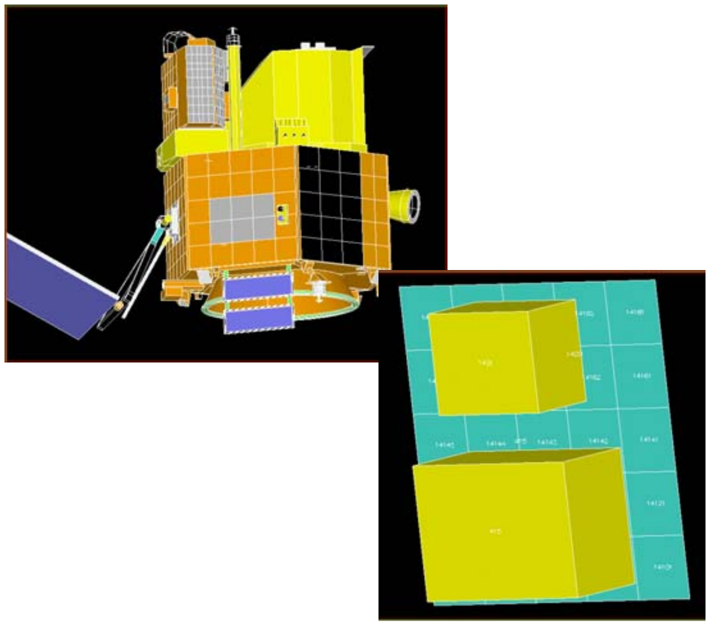

**Figure 9. Geometric Math Models and Thermal Math Model**

Bay 4 panel radiator design summary information is given in Table 2.

**Table 2. Bay 4 Carbon-Carbon Radiator Design Summary**

| PSE Box Dissipation | DCE | Standby | Safe |
|---|---|---|---|
| | 27.8W | 27.8W | 27.0W |
| LEISA/AC Box Dissipation | 16.3W | 0.0 | 0.0 |
| Heater Resistance | 75.7 ohms | | |
| Heater Power | 21V | 28V | 35V |
| | 5.8W | 10.4W | 16.2W |
| External Radiator Size | = 1408.59 sq cm<br>= 194 sq in. (12.0 x 16.25) | | |
| Thermal Design Description | Box is black anodize, chotherm at box/panel interface, AgTe radiator | | |

## 3.1 Thermal Ground Test Verification

Prior to spacecraft level testing, the CCR panel was subjected to four thermal vacuum cycles each consisting of a hot soak at 60°C for four hours and a cold soak at –20°C for four hours (Figure 10). A thermal balance test was performed at the completion of the thermal vacuum cycle test.

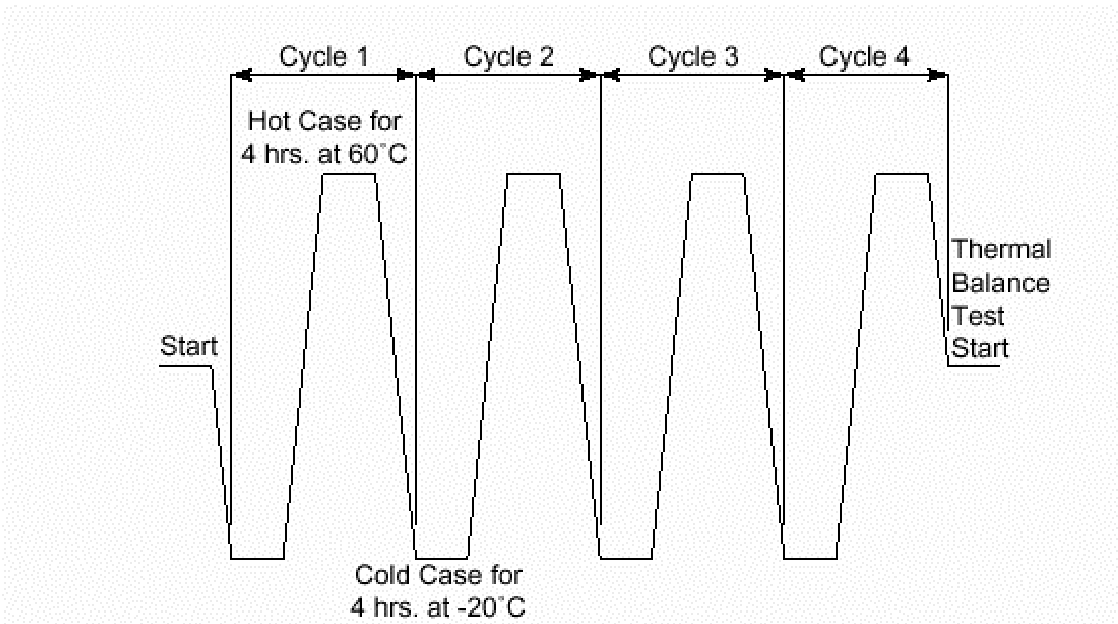


**Figure 10. Carbon-Carbon Radiator Panel Thermal Vacuum Cycle Test Profile**

Figure 11 shows an illustration of the CCR component level thermal vacuum/thermal balance test setup.

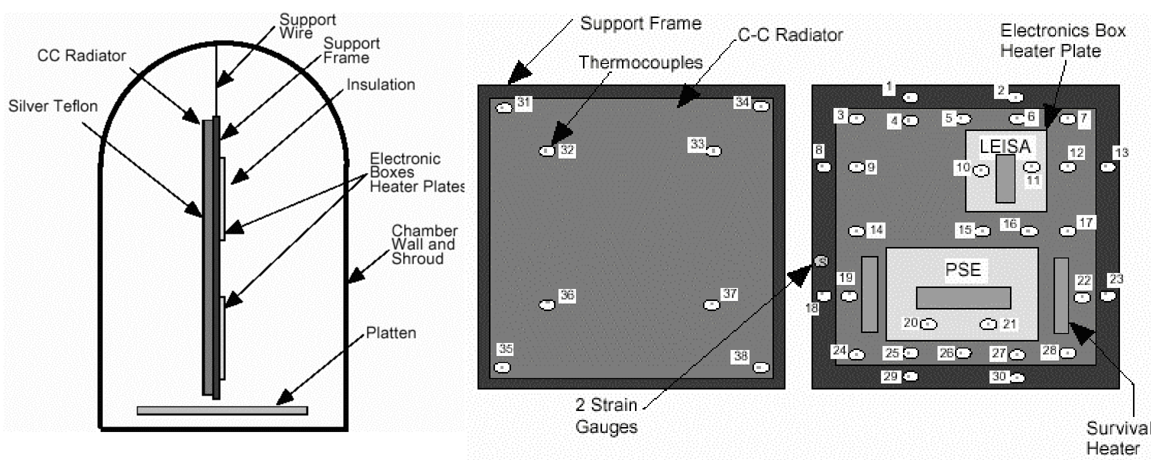


**Figure 11. Carbon-Carbon Radiator Panel Thermal Vacuum/Thermal Balance Test Setup**

The results of the component level thermal balance test are shown in Table 3. The cold case assumed 10 watts for the PSE and 0 watts for LEISA/AC and the hot case assumed 50 watts for the PSE and 30 watts for the LEISA/AC. The test results compared very well with the model analysis.

**Table 3. Carbon-Carbon Radiator Panel Thermal Balance Test Results**

| Location | Cold Case Actual | Cold Case Analysis Model | Hot Case Actual | Hot Case Analysis Model |
|---|---|---|---|---|
| TC15 | -9°C | -11°C | 28°C | 30°C |
| TC36 | -11°C | -13°C | 18°C | 18°C |

In addition to the component-level thermal vacuum/balance testing, the CCR also went through two spacecraft-level thermal vacuum tests. The first test included a comprehensive thermal balance test. Both tests had the CCR panel in the flight configuration. Results obtained during the spacecraft thermal balance test (HOT balance point) are shown in Figure 12. Thermal model test predictions and test results agreed within 1 degree and thereby verified the component thermal model.

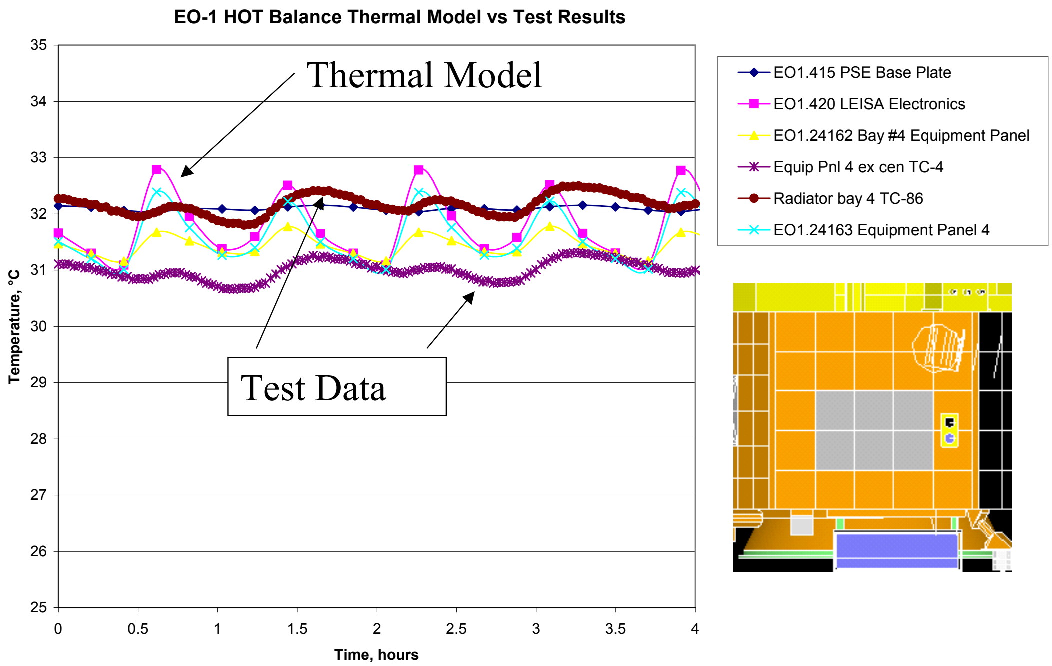


**Figure 12. EO-1 Spacecraft Thermal Balance Test Results**

## 3.2 Mechanical Ground Test Verification

During launch of the EO-1 spacecraft, the CCR was required to support the total (60 lb) weight of the two electronic boxes under an equivalent static load of 15G's. The panel was also required to react primary structural loads induced by the spacecraft bus. The first mode frequency of the panel and electronics box assembly, when attached to ground at the spacecraft mounting points, was to exceed 100 Hz.

The CCR, with electronics box assemblies, was sine burst tested to 18.8G (1.25 x 15G) at 22.5 Hz for 5 cycles in all three axes and was accompanied by pre and post low-level sine sweeps across the frequency range of 15-200 Hz. Mass mock-ups were designed to simulate the inertial properties of the electronics boxes and were attached to the panel during vibration testing. The panel was visually inspected and a non-destructive radiographic evaluation made after each test. The inspection examined inserts at the four corners of the panel as well as the corner inserts of the box mounting locations. The results of the inspections indicated no change to either the potting compound or the surrounding honeycomb core at those locations as a result of the tests. Consequently, the panel survived the sine burst loads, which indicated that there were positive margins on the acceleration loads. The first mode frequency of the test configuration was 89Hz. The first mode frequency predicted by finite element analysis of the test configuration was 93Hz. The analysis of the CCR and box assembly attached to ground predicted a first mode frequency of 112Hz. The model and test results were within 5%. These results verified the CCR finite element model and indicated that the first mode frequency of 112Hz, computed using the panel hard mounted at the spacecraft attachment points configuration, is reliable and satisfies the design requirement of 100 Hz or greater.

In flight, the CCR, which has a very low CTE, is attached to the EO-1 aluminum spacecraft bus, which has a high CTE. This differential in CTEs may result in substantial stresses as the on-orbit spacecraft temperature varies in the range of -10˚C to +50˚C. For flight qualification, the panel was analyzed for the range of -20˚C to +60˚C. The CCR was subjected to thermal cycling testing in accordance with Section 3.1, where an aluminum frame designed to simulate the stiffness of the spacecraft, was attached to the panel during each phase of testing. Strains were recorded during the thermal cycling. The thermal cycle testing served as an important structural test.

Dynamic analysis showed that the CCR met the stiffness requirement and stress analysis also showed positive margins for all specified load conditions. Radiographic inspection of critical insert locations, before and after testing, showed no change in the CCR microstructure, vibration sine sweeps showed no change in the CCR resonance characteristics, and visual inspection revealed no apparent damage. Therefore, the CCR was shown, through analysis and testing, to meet or exceed all launch and on-orbit performance requirements and was fully qualified for flight on the EO-1 spacecraft. Finite element and hand analyses were used to compute margins of safety for the radiator facesheets, honeycomb core, fasteners, and insert-potting compound. The margins of safety were determined to be positive in all cases for loads induced by both the 15G inertial load and 40˚C change in temperature. Test results showed that the panel successfully met the requirements for the EO-1 mission.

### 3.3 On-Orbit Thermal Test Validation

The CCR panel flight temperature measurements, shown in Figure 13, correlated extremely well with the predicted values obtained from the EO-1 spacecraft thermal model. The excellent correlation results can be attributed to the knowledge obtained during the spacecraft-level thermal vacuum test and understanding of the interfaces and power consumption of the flight electronics.

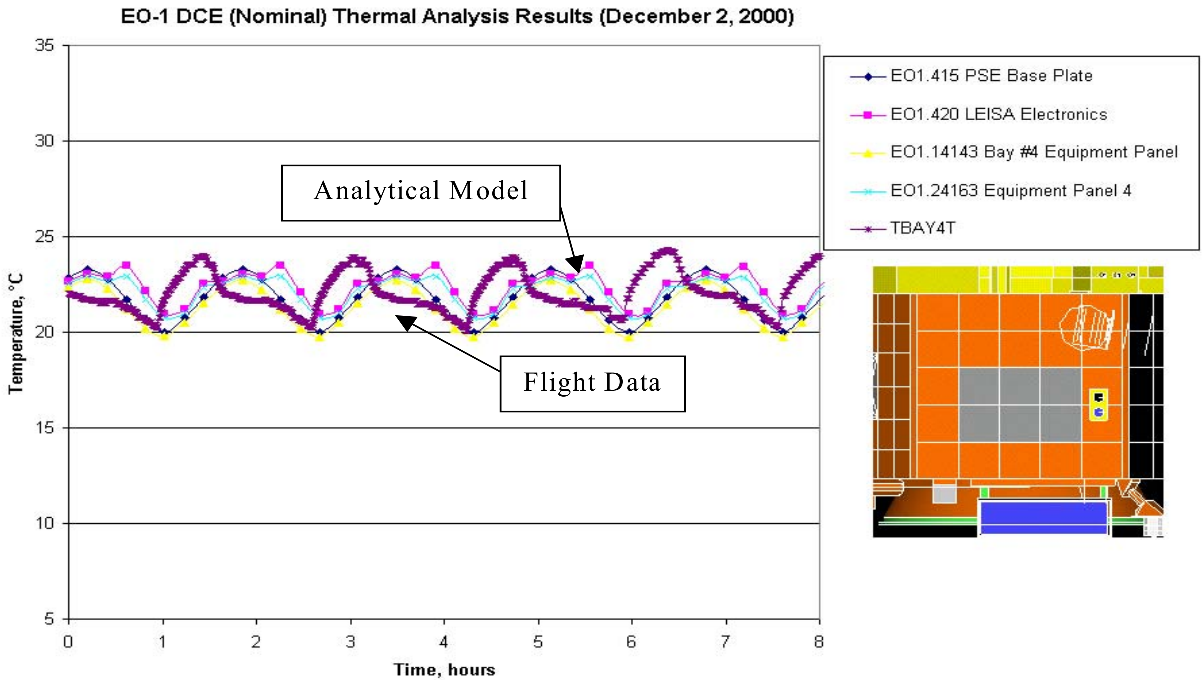


**Figure 13. Comparison of Flight Data with Analytical Model Data**

Analysis results were generated for four C-C 1N-plane thermal conductivity (k) values, k=200, 215, 230 and 245 W/m-K. Although the fidelity of the thermal model does not show a noticeable difference in the gradients for these four cases, using k=230 W/m-K produced the best correlation for the maximum and minimum temperatures for every thermistor temperature value on the C-C Radiator Panel. Therefore, the analytical results for the on-orbit thermal conductivity verified the pre-flight thermal conductivity value of 230 W/m-K.

The values shown in Table 4 were used in the final flight thermal model and produced excellent correlation between flight temperatures and thermal model predictions as shown in Figure 13.

**Table 4. Carbon-Carbon Radiator Conductance and Heat Transfer Coefficient Values**

| | **Pre-Flight / Experimental** | **Flight Analysis** |
|---|---|---|
| **k (horizontal)** | 230 W/m-K | 230 W/m-K |
| **k (vertical)** | 230 W/m-K | 230 W/m-K |
| **k (z direction)** | 30 W/m-K | 30 W/m-K |
| **h (honeycomb panel)*** | .0142 W/cm$^2$-K | .0076 W/cm$^2$-K |

*The pre-flight heat transfer coefficient (h), used to model the heat transfer through the honeycomb panel was based on empirical data for aluminum honeycomb panels of density 2.1 lb/ft$^3$. Correlation using flight data and analysis yields a heat transfer coefficient for CC face sheets approximately 50% less than the aluminum which can be attributed to the low (less than aluminum) through-thickness ($k_z$) thermal conductivity of the C-C facesheets.

Performance of the CCR panel was excellent throughout the mission. Review of the flight data for the first 12 months showed only a small change in temperature values. After 15 months on orbit, the EO-1 spacecraft continued to perform nominally. Table 5 contains the temperature data for the CCR panel thermistors one week after launch and approximately 12 months following launch. ***The temperatures represent a time-weighted average for a complete 24-hour day***. These flight data temperature readings show approximately a 1°C temperature rise for the CCR panel. This small temperature change can be attributed to optical property degradation of the silver Teflon radiator panel or possibly an increase in power. However, such a small change indicates the CCR panel on-orbit performance was excellent.

**Table 5. 12 Months Temperature Comparison (°C)**

| | TBAY4T | TRADCC2T | TRADCC3T | TRADCC4T | TRADCC5T | TRADCC6T |
|---|---|---|---|---|---|---|
| **11/28/2000** | 22.68 | 21.14 | 24.16 | 24.84 | 27.09 | 26.39 |
| **11/12/2001** | 23.72 | 21.88 | 25.23 | 26.04 | 28.33 | 27.87 |
| **DeltaT** | 1.04 | 0.74 | 1.07 | 1.2 | 1.24 | 1.48 |

In addition, Table 6 shows a comparison of flight temperatures for the hot orbital season (December 21, 2000) and the cold orbital season (June 21, 2001). ***The temperatures represent a time-weighted average for a complete 24-hour day***. The results are consistent with the change in environmental flux for the hot and cold orbit environments.

**Table 6. Hot / Cold Orbital Season Temperature Comparison (°C)**

| | TBAY4T | TRADCC2T | TRADCC3T | TRADCC4T | TRADCC5T | TRADCC6T |
|---|---|---|---|---|---|---|
| **12/21/2000** | 23.42 | 21.79 | 24.68 | 25.56 | 27.79 | 26.99 |
| **6/21/2001** | 21.73 | 20.1 | 23.31 | 23.97 | 26.2 | 25.67 |
| **DeltaT** | -1.69 | -1.69 | -1.37 | -1.59 | -1.59 | -1.32 |

The CCR panel temperature values for November 21, 2000 through November 12, 2001, are shown in Figures 14 through 18. The minimum and maximum temperatures were taken approximately once a week for each week following launch and the average temperature represents a time-weighted average for a complete 24-hour day taken approximately once a week for each week following launch.

There are labels on the figures indicating the approximate hot and cold orbital season. At points where the temperature data appears to increase a few degrees more than nominally usually indicates when a spacecraft maneuver was performed. For the most part there is little or no variation in temperature indicating that the CCR panel maintained excellent performance throughout the mission.

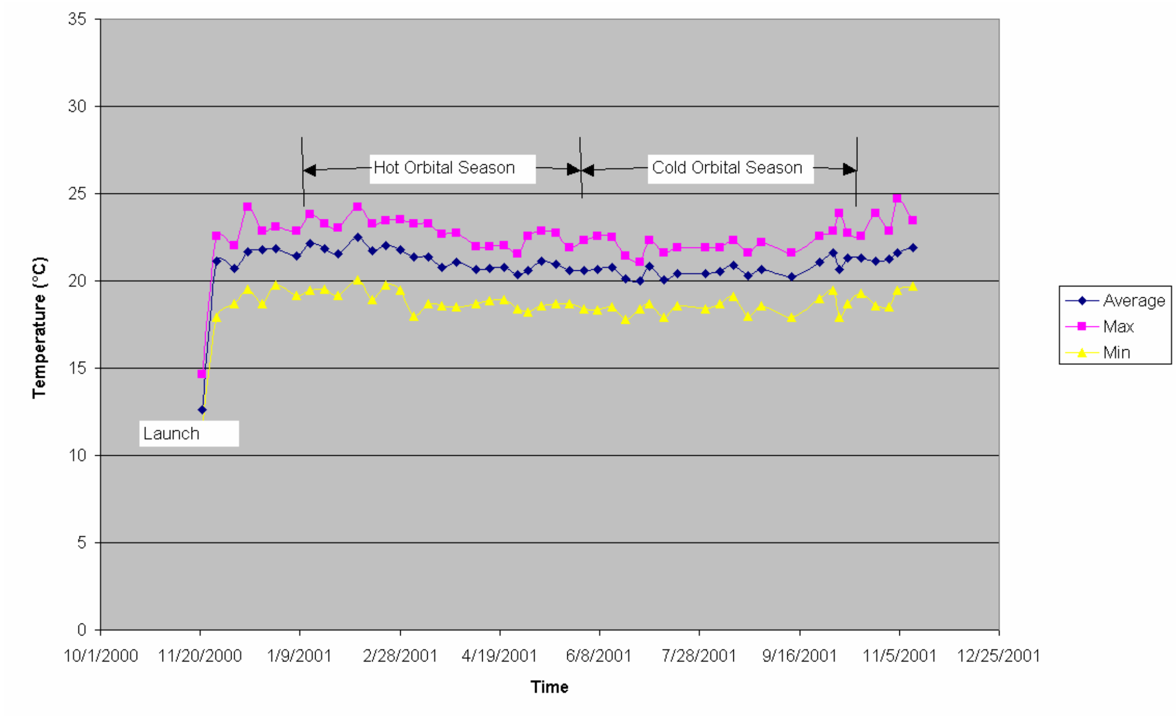


**Figure 14 Carbon-Carbon Radiator Panel Flight Temperatures (TRADCC2T)**

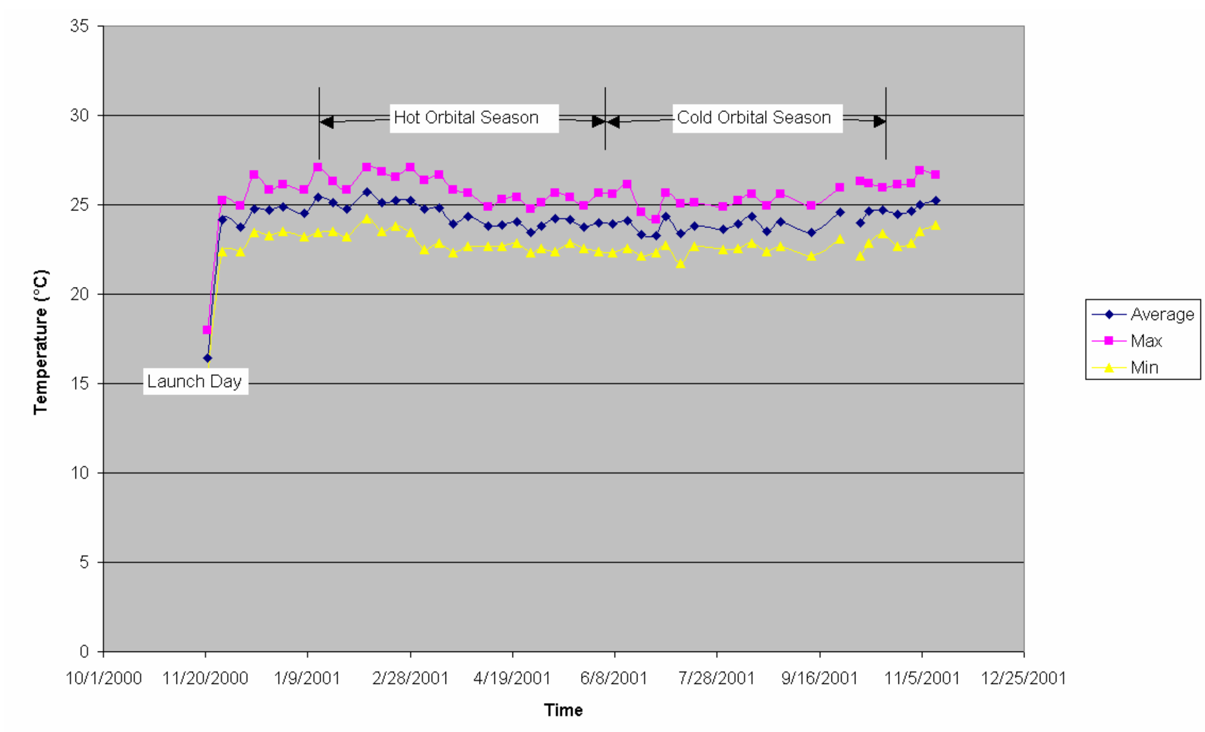


**Figure 15. Carbon-Carbon Radiator Panel Flight Temperatures (TRADCC3T)**

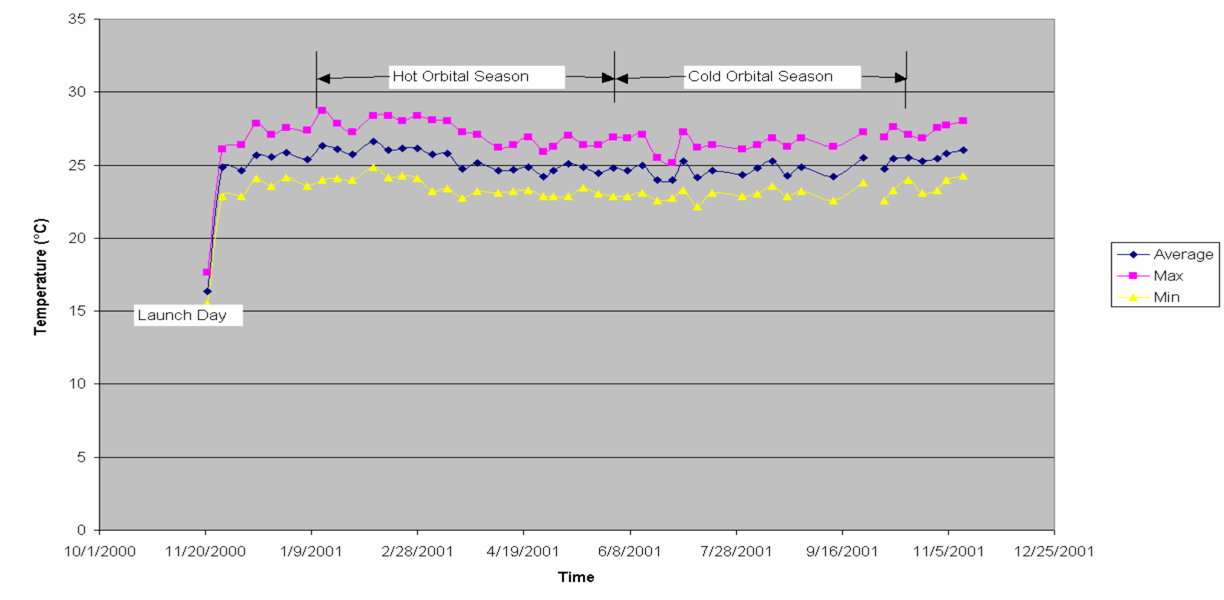


**Figure 16. Carbon-Carbon Radiator Panel Flight Temperatures (TRADCC4T)**

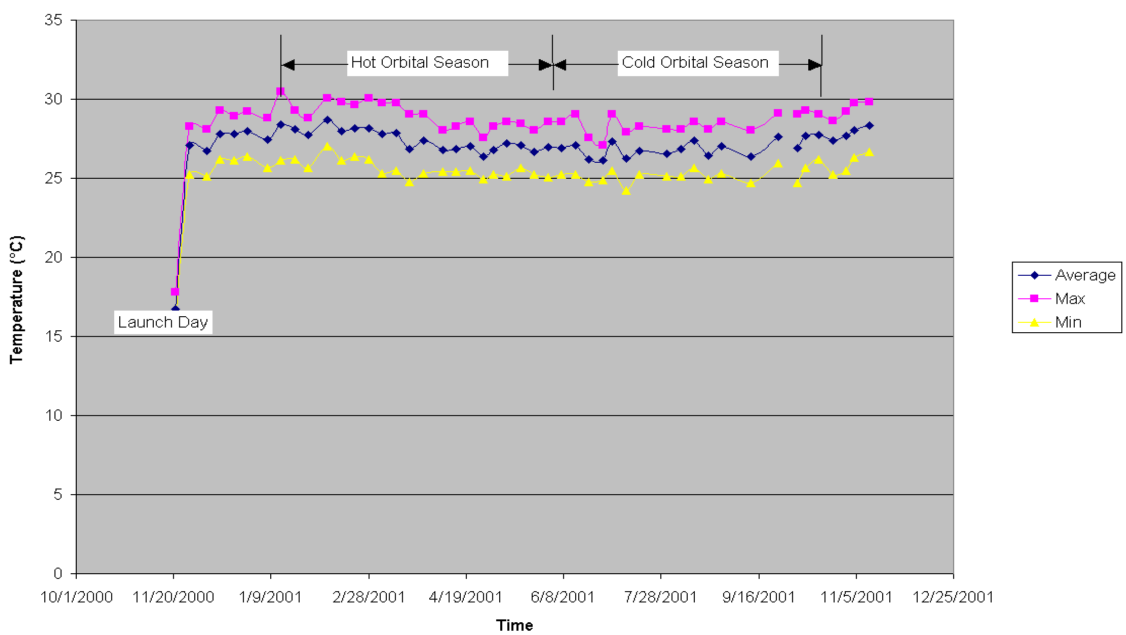


**Figure 17. Carbon-Carbon Radiator Panel Flight Temperatures (TRADCC5T)**

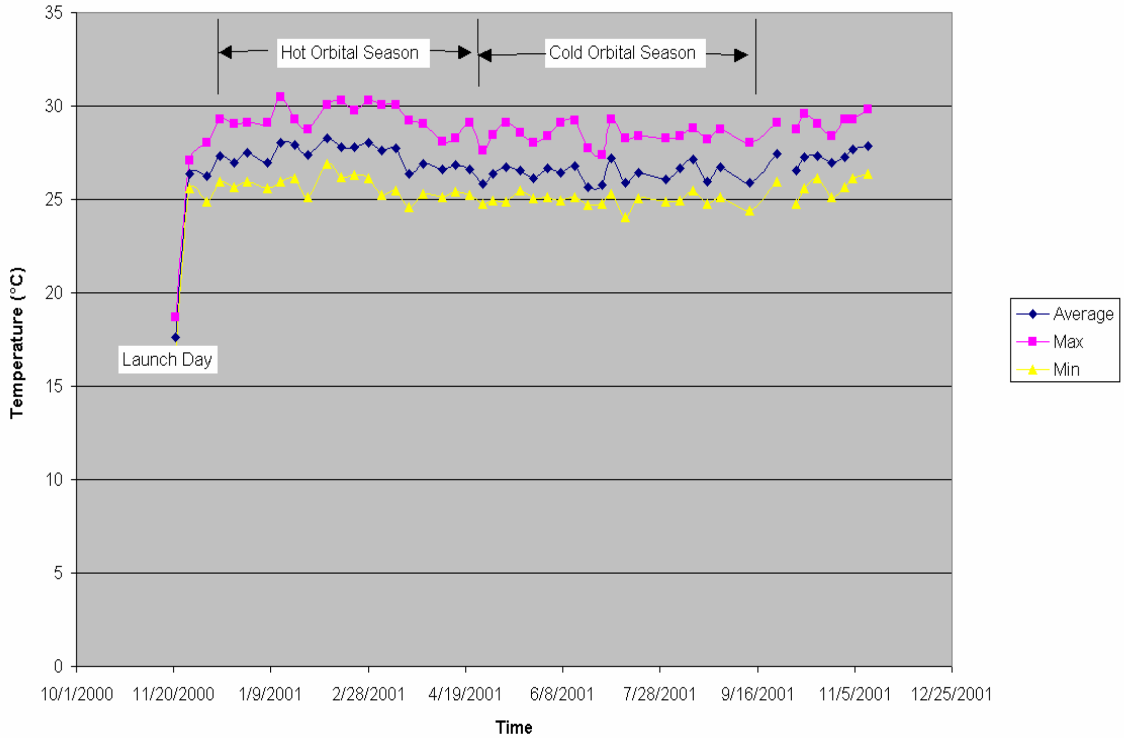


**Figure 18. Carbon-Carbon Radiator Panel Flight Temperatures (TRADCC6T)**

### 3.4 On-Orbit Usage Experience

The CCR panel met the mission requirements and its on-orbit performance was flawless. The mission requirements were to provide mechanical and thermal accommodations for the LEISA Atmospheric Corrector and EO-1 power supply electronics and to serve as a spacecraft primary structural member.

## 4. CURRENT APPLICATIONS AND NEW APPLICATIONS POSSIBILITIES

The success of the EO-1 CCR panel is complemented by some current applications and potential future applications. These include, but are not limited to:

- High Conductivity Optical Bench
- Nose cone of the space shuttle
- Re-entry vehicles that pierce the atmosphere twice as it enters space and returns to Earth
- Aircraft brakes
- Wing leading edges
- Engine nozzles

## 5. FUTURE MISSIONS INFUSION OPPORTUNITIES

- Possible use of C-C foam as a low weight, low CTE mirror/optical bench substrate.
- Use of CCR panels to replace aluminum radiator panels.
- Use of CCR panels to replace entire heat pipe panels.

- C-C material may eliminate need for thermal doubler plates.
- Use of CCR where its tailorable properties are needed to satisfy design requirements.

## 6. LESSONS LEARNED

- The C-C Radiator was a success and proved that the technology can work to reduce spacecraft weight.
- It has been demonstrated that sandwich honeycomb composite panels, with C-C facesheet material, are a viable option for accommodating both thermally and structurally demanding application requirements into a single design.
- C-C has a niche, especially for high temperatures.
- C-C still needs further development
  - Reduction in fabrication time and cost - high conductivity "traditional" composites are more competitive
  - CTE interface issues with heat pipes
- Having redundant hardware available is good idea - spare panel was flown since the flight unit was inadvertently damaged.
- Informal inter-agency partnership can be highly effective - CSRP was a success.

## 7. CONTACT INFORMATION

| | |
|---|---|
| Nicholas M. Teti | C. Dan Butler |
| Swales Aerospace | NASA/GSFC |
| 5050 Powder Mill Road | Greenbelt, MD 20771 |
| Beltsville, MD 20705 | 301-286-3478 |
| 301-902-4100 | dan.butler@gsfc.nasa.gov |
| nteti@swales.com | |

## 8. SUMMARY

The thermal model results correlate very well with the EO-1 flight data. A thermal conductivity of 230 W/m-K predicts the best correlation for the maximum and minimum temperatures for every thermistor temperature value on the CCR Panel. Therefore, the analytical results for the on-orbit thermal conductivity validate the pre-flight thermal conductivity value of 230 W/m-K. Performance of the CCR panel was excellent throughout the mission. Review of the flight data for the first 12 months on-orbit showed only a small change in temperature values (<1.5°C rise). The plan is to continue to evaluate the performance of the CCR panel for the entire EO-1 extended mission, re-evaluate the data and provide an updated report to NASA/GSFC.

## 9. CONCLUSIONS

The CCR was a success and proved that the technology can work and should continue to find space applications, especially for high temperature thermal management applications and for applications that nclude accommodating demanding structural design requirements. However, in an era when faster, better, cheaper is still a key focus for most space related projects, the C-C process still needs further development to obtain ways to reduce fabrication time and cost.

# 10. TECHNICAL REFERENCES

[1] Jonathan Kuhn, Steve Benner, Dan Butler and Eric Silk, "Thermal and Mechanical Performance of a Carbon/Carbon Composite Spacecraft Radiator," Proceedings SPIE Conference, Vol. 3786, Optomechanical Engineering and Vibration Control, pp. 162-178, 09/1999.

[2] Timothy Anderl, "Unique Partnership Yields Spacecraft Improved Radiator Panel," *United States Air Force News Release,* Aeronautical Systems Center Public Affairs, Wright-Patterson AFB, OH 45433.